

UNCLASSIFIED

AD NUMBER

AD866347

LIMITATION CHANGES

TO:

Approved for public release; distribution is unlimited.

FROM:

**Distribution: Further dissemination only as directed by Army Aviation Systems Command,
Attn: AMSAV-R-F, St. Louis, MO 63166, FEB 1970,
or higher DoD authority.**

AUTHORITY

USAAVSCOM ltr, 12 Nov 1973

THIS PAGE IS UNCLASSIFIED

AD 866347

EDS FILE COPY



20
AD
ROUTE PROJECT NO.
USAAVSCOM PROJECT NO. 68-02
USAASSTA PROJECT NO. 68-02

CH-47B HEIGHT-VELOCITY EVALUATION

FINAL REPORT

ALAN L. HOFFMAN
CPT, ORD C
US ARMY
PROJECT ENGINEER

WAYNE B. DAVIS
MAJ, CE
US ARMY
PROJECT PILOT

JOHN J. SHAPLEY, JR.
PROJECT OFFICER

FEBRUARY 1970

DISTRIBUTION

This document may be further distributed by any holder only with specific prior approval obtained through the CG, USAAVSCOM, ATTN: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166.

US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

(see 1... 1973)

DISCLAIMER NOTICE

The findings in this report are not to be construed as an official Department of the Army position unless so designated by other authorized documents.

DDC AVAILABILITY NOTICE

[REDACTED]

REPRODUCTION LIMITATIONS

Reproduction of this document in whole or in part is prohibited except with permission obtained through the Commanding General, USAAVS-COM, ATTN: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166. DDC is authorized to reproduce the document for United States Government purposes.

DISPOSITION INSTRUCTIONS

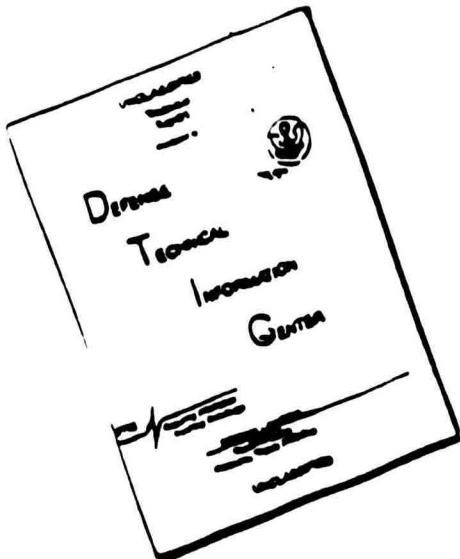
Destroy this report when it is no longer needed. Do not return it to the originator.

TRADE NAMES

The use of trade names in this report does not constitute an official endorsement or approval of the use of the commercial hardware and software.

7
5

DISCLAIMER NOTICE



**THIS DOCUMENT IS BEST
QUALITY AVAILABLE. THE COPY
FURNISHED TO DTIC CONTAINED
A SIGNIFICANT NUMBER OF
PAGES WHICH DO NOT
REPRODUCE LEGIBLY.**

ROUTE PROJECT NO.
USA AVSCOM PROJECT NO. 68-02
USA AASTA PROJECT NO. 68-02

CH-47B HEIGHT-VELOCITY EVALUATION

FINAL REPORT

TCC
LTS

ALAN L. HOFFMAN
CPT, ORD C
US ARMY
PROJECT ENGINEER

WAYNE B. DAVIS
MAJ, CE
US ARMY
PROJECT PILOT

JOHN J. SHAPLEY, JR.
PROJECT OFFICER

FEBRUARY 1970

DISTRIBUTION

This document may be further distributed by any holder only with
specific prior approval obtained through the CG, USAAVSCOM, ATTN:
AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166.

US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

ABSTRACT

The CH-47B height-velocity testing was conducted at Bakersfield and Lake Tahoe, California, during the period 21 October 1968 to 7 February 1969. The program was a joint effort by the US Army Aviation Systems Test Activity (USAASTA) and Vertol Division of The Boeing Company (Boeing-Vertol). Boeing-Vertol personnel developed the recommended diagrams which were then evaluated by USAASTA personnel. A modified version of the power deficiency parameter method of prediction provides an accuracy similar to that with which the tests can be conducted and data recorded. The resulting height-velocity diagrams were obtained with a specialized technique and may not be suitable for inclusion in the operator's manual. Thus, there is a need to define a suitable pilot technique, make the necessary calculations and accomplish the necessary verification testing.

TABLE OF CONTENTS

	<u>Page</u>
INTRODUCTION	1
Background	1
Objectives	1
Description	1
Scope of Test	1
Method of Test	2
Chronology	3
RESULTS AND DISCUSSION	4
General	4
Analysis of the Power Deficiency Parameter	5
Height-Velocity Predictions	9
Analysis of Test Results	11
Autorotational Landing	13
CONCLUSIONS	15
RECOMMENDATIONS	18
APPENDIXES	17
I. References	17
II. Test Data	18
III. Distribution	20

INTRODUCTION

BACKGROUND

1. The initial CH-47A testing conducted by the Air Force Flight Test Center was terminated after generation of one height-velocity (H-V) curve (28,000-pound gross weight (grwt) at a 2630-foot density altitude (H_D)) (ref 8, app I). Gross weight and altitude effects were not evaluated. A further development of the CH-47B and CH-47C models incorporated a modified rotor system and increased the gross weight limitation. The Vertol Division of The Boeing Company (Boeing-Vertol) derived a method for predicting and defining the H-V performance in terms of a power deficiency parameter (PDP). This parameter is essentially based on power required for hover, power available and energy requirements during descent and landing. Boeing-Vertol submitted a proposal to validate the PDP method through flight testing and at the same time develop H-V curves for the operator's manual. The contract included the test procedures as specified in reference 3, appendix I. This essentially required Boeing-Vertol to establish the H-V curves. The US Army Aviation Systems Test Activity (USAASTA) was directed in reference 1 to validate the results for incorporation in the operator's manual.

OBJECTIVES

2. The test objectives were to evaluate the Boeing-Vertol method of predicting and calculating H-V performance, obtain representative curves for the operator's handbook and investigate performance characteristics with respect to altitude and gross weight.

DESCRIPTION

3. The initial portion of the program was conducted with a CH-47B helicopter (S/N 66-19100). Following an accident, the program was completed with a CH-47C (S/N 68-15820). With the exception of test equipment discussed herein, both aircraft were of standard configuration and are described in detail in references 4 and 7, appendix I.

SCOPE OF TEST

4. The scope of test was rather comprehensive in that a large range of power margins was tested. This variation included both gross

weight and power effects. The testing was accomplished at two altitudes to evaluate performance changes versus altitude. Rotor speed was 225 rpm for weights below 37,500 pounds and 230 rpm for heavier weights. Delay time as specified in the contract was a minimum of 1 second for all tests, and the pilot's technique was "optimum". This optimum technique was developed during the build-up procedure and was based on quantitative data and analysis of the results, as well as the pilot's qualitative comments. General ground rules were that the maximum H-V condition would not be used for the tests, but realistic, normal conditions would be demonstrated. This was in the interest of good flight safety practices and to provide data for normal operations.

METHOD OF TEST

5. The test instrumentation included real time telemetry for critical parameters, such as gear loads and rate of descent (R/D). Maximum allowable (red-line) values for all critical parameters were established prior to starting the tests. The initial portion of the test was accomplished with a rather pragmatic approach in that only limited engineering effort had been performed. The pilot incrementally decreased the margin on a given flight parameter, such as airspeed or altitude, until the qualitative or quantitative limit was reached. During the course of the program, the initial engineering approach was found to be inadequate and additional analysis was accomplished which resulted in predictions that were based on the flight test data gathered to this point. These were used to further increase the safety and to advise the test team of critical conditions.

6. The autorotational entry test procedure was to stabilize at the specified condition and retard one engine to ground idle. The engines were rigged to provide the specification power available for the ambient test conditions. Following this, the pilot would delay 1 second and then use the cyclic pitch attitude control to achieve the desired flight path and the collective to manage rotor speed. The contractor's technique stipulated that the pilot maintain rotor speed at 215 to 220 rpm which assured maximum power of the remaining engine. The ground idle speed was adjusted so that no power was being derived from the retarded engine. The flare was also controlled by maintaining the rotor speed below 220 rpm (above which an engine torque reduction occurs). Following touch-down, aerodynamic and wheel braking were used to stop the aircraft.

CHRONOLOGY

7. The chronology of the test program is presented below:

Test directive received	10 August 1968
Test aircraft received	20 August 1968
Tests started	21 October 1968
Tests terminated by accident	26 October 1968
Replacement aircraft received	6 November 1968
Tests continued	7 December 1968
Tests completed	7 February 1969
Draft report submitted	November 1969

RESULTS AND DISCUSSION

GENERAL

8. Altitude effects were ascertained by conducting the tests at approximate 500- and 6000-foot density altitudes. The H-V testing in both cases can be divided into three general areas:

- a. Low height, low airspeed.
- b. Medium height, high airspeed.
- c. High height, low airspeed.

9. The first area encompasses in-ground-effect (IGE) hover and IGE level flight and acceleration. The second area encompasses slow level flight, accelerating flight and rotation, plus high-speed level flight where the limiting factor is the forward landing speed limit of the gear. The third area is out-of-ground-effect (OGE) hover.

10. The build-up flights at the 500-foot H_p were conducted at 33,000 and 37,000 pounds and led to the final testing at 40,000 pounds. In all cases, one engine was retarded to ground idle to begin the partial power descent. The remaining engine's power output was increased in an attempt to maintain governed rotor speed. The maximum power of the remaining engine was previously set to yield the desired power deficiency parameter which is the available shaft horsepower (shp) minus the required shp for hover divided by the gross weight (W) of the aircraft. Equation 1 shows how PDP is derived:

$$PDP = \frac{SHP_{avail} - SHP_{rqrd \text{ for OGE hover}}}{W} \quad (1)$$

11. This PDP is based on power required for OGE hover and gives a measure of the expected vertical steady state R/D in a vertical OGE descent:

$$\begin{aligned} R/D &= \frac{550K_p}{W} (SHP_{rqrd \text{ for OGE hover}} - SHP_{avail}) \\ &= 550K_p (-PDP) \quad (\text{fps}) \end{aligned} \quad (2)$$

12. Thus, power losses at OGE hover with different gross weights and density altitudes but the same PDP and referred gwt (W/δ) should result in the same steady state R/D; this is assuming that the climb power correction factor (K_p) does not change with these variables based on these two related effects. However, this does not mean that the total H-V performance will be the same for the two cases even if only vertical descents are considered.

ANALYSIS OF THE POWER DEFICIENCY PARAMETER

13. Theoretically, as shown above, two flights conducted at different weights and altitudes but the same PDP would have the same steady state vertical OGE R/D. This is not true for nonvertical descents:

$$R/D = \frac{550K_p}{W} (\text{SHP}_{\text{rqrd}} - \text{SHP}_{\text{avail}}) = 550K_p$$

$$[(-PDP) - \frac{1}{W} (\text{SHP}_{\text{rqrd for OGE hover}} - \text{SHP}_{\text{rqrd}})] \quad (3)$$

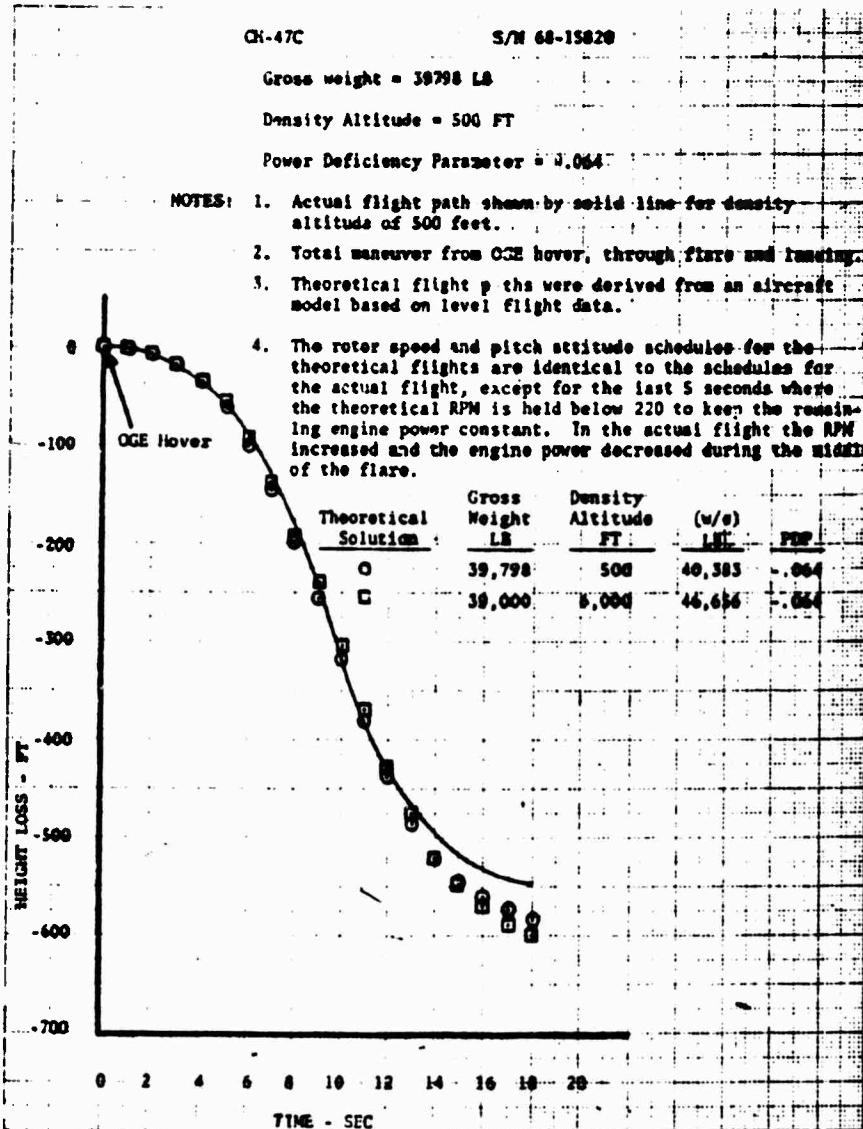
14. If the two flights are flown at different values of W/σ , they will have different steady state nonvertical rates of descent; unless, for a given velocity, the SHP required for OGE hover minus the SHP required for that velocity is proportional to W . In equation form:

$$\left[\frac{\text{SHP}_{\text{rqrd for OGE hover}} - \text{SHP}_{\text{rqrd}}}{\sigma} \right] = \frac{W}{\sigma} \quad (4)$$

Depending on the velocities and W/σ , either the larger or the smaller referred gross weight case may have a greater steady state R/D.

15. By its very nature, a critical H-V maneuver does not consist of a steady state condition. This further nullifies the validity of a PDP method based on both vertical descent and steady state conditions. A typical H-V flight trajectory consists of an initial downward acceleration as thrust decreases due to power loss and a subsequent deceleration during the flare portion as thrust is increased. Figure A shows an actual flight trajectory flown at the 500-foot H_p and a PDP of -0.064. Two computer solutions are also shown for predicted theoretical flight paths at the same PDP for the 500- and 6000-foot H_p . The computer solutions are based on level-flight data which were used to determine the CH-47 power-required

FIGURE A



parameters. The pilot inputs (as evidenced by rotor speed and pitch attitude schedules) were the same as for the actual flight except during the latter portion of the flare. For the last 5 seconds of the actual flight, the rotor speed increased above 220 rpm, and the remaining engine torque dropped rapidly. In the theoretical solutions, the rotor speed was maintained at or below 220 rpm, and the remaining engine power was held constant.

16. The close agreement of the theoretical flight path with the actual flight path, except during the flare, attests to the usefulness of the prediction method. The CH-47 level-flight data does not reveal the increased rotor effectiveness during a flare. This effect can be included in more refined models by incorporating actual H-V data. Ground effect is very important but was not included since the testing was conducted completely OGE.

17. The good applicability of level-flight parameters to the highly transient H-V conditions is fortuitous, although it is to be expected that some changes may affect two theoretical flights. While the close agreement between the actual and theoretical runs may be accidental, the differential effect of the change in referred grwt between the two theoretical flights shown in figure A should be in close agreement with the change occurring during actual flight tests. This is true even for the flare portion where the absolute performance is underestimated. For the first 4 seconds of the OGE hover test as shown in figure A, the two theoretical curves agree since the PDP method is based on OGE hover characteristics and since there is no great deviation from steady state conditions. From 4 to 10 seconds during the nose-down portion, the R/D increase is slightly greater for the lower H_p (also lower referred grwt) case. During the flare, the R/D is not reduced as quickly as it was during the higher referred grwt, higher H_p case.

18. The phenomenon (where the H-V performance for two flights at the same PDP, but different gross weight, is better for the low referred grwt case during the flare portion) can be explained by equations 5, 6 and 7. The available horsepower in any given situation is equal to that supplied by the remaining engine power plus that gained from potential, kinetic and rotor inertia energy as shown in equation 5:

$$SHP = SHP_{\text{engine}} + \frac{1}{550} (R/D \times W - M \dot{V} \cdot \dot{a} - I \ddot{\Omega}) \quad (5)$$

Solving for the SHP_{avail} in equation 1 and substituting for SHP_{eng} in equation 5, the following equation is obtained:

$$SHP = SHP_{\text{reqd for OGE hover}} + W$$

$$\left[\frac{R/D}{550} - \frac{1}{550g} \hat{V} \cdot \hat{a} - (-PDP) \right] - \frac{I}{550} \dot{\Omega} \quad (6)$$

By dividing equation 6 by ρAV_b^3 [where the rotor tip speed (V_b) equals ΩR] and defining a rotor inertia coefficient (C_I) as $I/\rho AR^3$ with C_p and C_T defined in the normal manner, equation 7 is produced:

$$C_p (C_T, \mu) - C_p (C_W, 0) = C_W$$

$$\left[\frac{R/D}{550V_b} - \frac{\hat{V} \cdot \hat{a}}{550V_b g} - \left(\frac{-PDP}{V_b} \right) \right] - \left(\frac{C_I}{550} \right) \left(\frac{\dot{\Omega}}{\Omega^2} \right) \quad (7)$$

19. Ignoring the last term in equation 7 for the moment, for two flights with identical values of PDP, identical velocities, accelerations and rotor speeds, the only differences between the flights will be the OGE hover power required and the weight coefficient (C_W). C_W is equivalent to the thrust coefficient (C_T) when T equals W . For a given velocity, power required increases nonlinearly with thrust (a power coefficient (C_p) versus C_T curve for constant advance ratio (μ) equals V/V_b is concave upward), and it will take a larger increase in C_p to produce the same increase in C_T for an initially higher C_T level. Since the excess C_p from equation 7 is proportional to C_W for any set of conditions, the increase in C_T that can be produced will not be proportional to C_W . Phrasing it differently: For the same percentage increase in C_p , the percentage increase in C_T will be less for the higher C_W [or higher $C_p (C_W, 0)$] case. Thus, flare for higher C_W flights will be less effective (T/W will be lower) than flares for lower C_W flights at the same value of PDP. Likewise, percentage decreases in C_T will also be less for the higher C_W case; the low-thrust (nose-down) portions will result in higher values of T/W than low C_W cases; and the rates of descent will not increase as rapidly. This behavior is shown in figure A and is not so apparent for the OGE hover point since the flare portion, where most of the deviation occurs, is only a small part of the flight. This deviation would be important for a low-altitude point where the flare portion is the dominant feature. The PDP method based on an OGE hover would not be accurate even for steady conditions with nonzero horizontal speeds for the same reasons as discussed in paragraph 13.

20. One other effect is evident (equation 7) when the rotor speed is not constant and rotor energy is available. The rotor energy is not proportional to C_W or W but depends on the rotor speed and moment of inertia (I). The greater the aircraft weight, the less effective the fixed rotor energy will be in developing normal acceleration. It is easier to build rotor energy at a heavy grwt, but since the touchdown rotor speed is generally lower than the rotor speed at throttle chop, the overall performance gain is less for a heavier grwt. This also detracts from the validity of the PDP method for predicting H-V performance. The following section shows a more detailed and accurate method which can be improved upon by actual test data.

HEIGHT-VELOCITY PREDICTIONS

21. During the initial testing at low altitude (500-foot H_D), the aircraft was damaged while landing from a high-altitude hover condition. Review of the data and consideration of the prevailing atmospheric conditions revealed several noteworthy items. Wind conditions were being recorded at approximately 10 feet above the ground while the aircraft was at several hundred feet. Thus, with wind gradients and shear not accounted for, the recorded atmospheric data may not be representative of the aircraft's environment. The previous hover point (at a higher height) also indicated that at power loss the aircraft was moving forward and in a slight climb; whereas, for the point in question, there was no forward motion and a small R/D. The maneuver was being monitored in real time on the telemetry screen, but there was insufficient time to recognize the warnings, evaluate the results and warn the pilot. It is also significant that the procedure was to decrease the hover height in constant increments of 50 feet on successive points rather than varying the increment on the basis of criticality of the preceding point.

22. Evaluation of these factors indicated that prior to continuing the tests a change in some of the basic ground rules was needed and performance predictions would considerably increase flight safety. A rigorous treatment of the H-V maneuver was beyond the time and resources available for the program; however, it was decided to incorporate hover and level-flight characteristics data into an energy analysis that would approximate the actual test performance obtained up to the time of the accident.

23. When relating steady state or nonsteady state R/D to power, an improvement can be made on a PDP method by using the level-flight power required rather than the OGE hover power for a particular velocity. The same arguments as given above will apply to the

non-steady state portion. One method for making H-V predictions is to use level-flight data to arrive at the power required for any given combination of thrust, rotor speed and velocity. This can be done directly by using level-flight power-required curves or nondimensional C_p , C_T or u curves. The method used to compute the flight paths as shown in figure A and to predict H-V performance prior to testing uses the level-flight data to define the effective rotor area (A_{eff}) plus three aircraft power parameters: the parasite drag coefficient (C_d), the rotor profile power drag coefficient (C_{dr}) and the C_T . Using these power parameters has an advantage over the use of level-flight curves in that predictions can be made for attainable thrust levels for given conditions without using successive iterations. Also, the computing technique is quicker, simpler and less data storage is required to define the model. In addition, the effect of the changing air flow across the rotor blades due to pitch-overs and flares or rates of climb and descent can be accounted for in the induced power term.

24. Assuming a given rotor speed and pitch-attitude profile, the available power can be computed starting at some initial point. It will consist of the remaining engine power plus rate of change in potential, kinetic and rotor energy. The resultant thrust can then be calculated, and the accelerations can be determined by the pitch-attitude profile. From these acceleration computations, the actual flight path can be determined by successive steps. It is assumed that the pilot can fly the given pitch-attitude and rotor-speed profiles by manipulating his cyclic and collective controls. Actually, the rotor/pitch profile rather than the rotor-speed profile of the aircraft is desired. They are related through the rotor-flap angle, pitch acceleration, aircraft center of gravity (cg) and moment of inertia.

25. Two effects will detract from the accuracy of this method. At low speeds, IGE will improve performance. This can be compensated by increasing A_{eff} as a function of airspeed and determining the correct test height experimentally. The second effect is: Even when out of ground effect, A_{eff} is not constant. In level flight, it varies with airspeed, but the effect is not so great as to prevent accurate predictions using an average value. The effective area may vary considerably with angle of attack, especially in a tandem rotor aircraft. In a flare, the actual performance of the CH-47 can be better than predicted using a constant A_{eff} . This effect along with many small effects due to rates and accelerations can only be determined by detailed flight testing.

26. One beneficial aspect of ground effect and additional flare effectiveness is that they both increase the performance of the aircraft. Thus, all predictions based on pure level-flight data

will be conservative as shown in figure A. However, they will be extremely useful in predicting the effects of small changes in entry attitude or pilot techniques. For example, the effect of forward velocity on an OGE hover would be clearly evident, and the importance of monitoring or considering this factor would be recognized before actual flight tests were begun.

ANALYSIS OF TEST RESULTS

27. The contractor's test results are presented in reference 9, appendix I, from which figures 1 through 9, appendix II, were obtained. Following the contractor's tests, USAASTA evaluated the recommended H-V curves. The most critical factors were the reduction of forward speed at touchdown and the minimization of ground roll distance. The high-speed, low-height condition was least critical and is the most common technique. The high hover is not hazardous so long as the proper technique is used; however, this is the most unnatural to the pilot and may create excessive control problems. There was an unacceptable altimeter error in a hover. During one hover condition, the cockpit indicators both showed a height of 500 feet while the measured value was between 250 and 300 feet.

28. It was noted that the contractor and Army test techniques were somewhat different. Both techniques were similar during the initial portion where the nose of the aircraft was lowered to gain or maintain airspeed and minimize required rotor power. The contractor technique utilized a mild flare while preserving full torque from the remaining engine by holding rotor speed near 215 rpm. The touchdown was made in a nose-high attitude which was maintained to provide aerodynamic braking during the ground roll. The Army technique allowed the aircraft to flare more sharply and let rotor speed increase. The aircraft was then leveled prior to touchdown which was accomplished on all four landing gear. Wheel braking alone was used to stop the ground roll. This latter technique consistently produced lower touchdown speeds as the sharp flare effectively slowed the helicopter and increased rotor energy. Individual gear loads can be lowered by distributing the loads on all four gear as opposed to the aft gear only. It is significant that when rotor speed exceeds 220 rpm, approximately 5 seconds are needed for the torque to decrease to a minimum. Thus, engine power is not lost instantaneously, and the greater rotor energy acts as an aid during the collective application to cushion landing. The net difference between these two techniques is that the Army method considerably reduces the forward glide distance from 100 feet to touchdown and decreases landing speed. Typical landing speeds and distances are shown in figure 1, appendix II. These data show that for other than hover, zero touchdown speed is virtually impossible, and there will be some ground roll distance. At SL and the

40,000-pound g-wt, the minimum speed and ground roll distance were approximately 25 knots and 225 feet. As performance decreases (higher weight, higher altitude and lower power), both landing speed and glide distance increase significantly.

29. As expected, increasing gross weight or decreasing power reduces the capability to make a safe landing. The change is approximately the same for all airspeeds. The area near the "nose" of the curve was investigated from a level-flight entry condition and from a rotation-to-climb condition. There was concern over which was the more critical situation. The test data show no significant difference. It is also interesting to note that the "nose" does not exist for the predicted data shown in figures 5.23 and 5.25, reference 9, appendix I. Evidently, the upward momentum during rotation and climb adequately compensates for the higher engine power utilized and higher rotor power required after the power loss.

30. Following the aircraft accident, the program was delayed until the winter months, and the test-site average H_d was lowered to 6000 feet. To evaluate altitude effects, the PDP values at the high-altitude tests were the same as for the low-altitude tests. As before, the engines were adjusted to standard power, and the gross weight was adjusted to arrive at the proper PDP. Analytical calculations were used to predict the performance to increase flight safety and provide information as to the critical areas to be investigated.

31. The light-weight tests at the same PDP produced results similar to those obtained during the heavy-weight tests at SL. These results were in agreement with the predicted performance. A comparison of the data obtained is shown in figure 2, appendix II. The large amount of scatter is somewhat misleading in that each point shown does not have the same degree or criticality. Apparently, the modified PDP method discussed in reference 9, appendix I, can be used to predict performance to an accuracy equivalent only to that of the test data. There were no changes in pilot technique or any differentiating characteristics. The preliminary calculations indicated that the heavy gwt condition could be critical with respect to performance and technique. The specific criterion was that airspeed be maintained at a relatively high level unless the aircraft was near the ground; ie, the flight safety corridor was narrowed and deceleration and descent relationships were given more importance. As a result of the analysis, the anticipated safe corridor was established at a higher speed than it might otherwise have been. The high-speed portion was also modified to include a height variation with airspeed in the range from 50 to 90 knots. The test results agreed well with the anticipated

performance. As was the case at low altitude, the contractor results were evaluated by USAASTA, and the evaluation was similar.

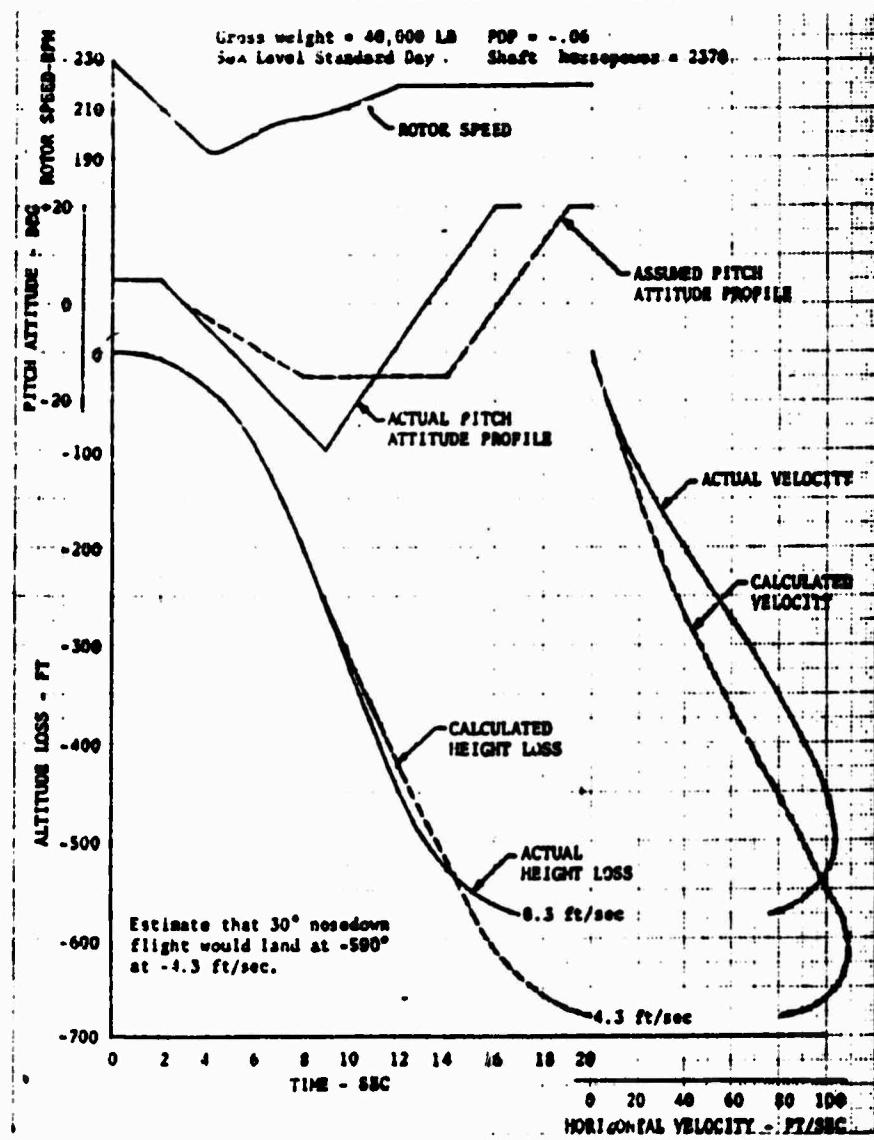
32. The test philosophy was to produce data which was realistic with respect to operational conditions. As such, the test results do not show the maximum capability of the aircraft. This realistic approach actually produces more valuable data for Army evaluation, while considerably reducing the test hazards and costs. The H-V curves developed during the tests are shown in figures 3 through 8, appendix II. These performance curves and the technique required have been presented to members of the US Army Aviation School at Fort Tucker, Alabama, for review and comment. These opinions are of particular value because of the pilot's experience in operational aspects and overall knowledge of Army aviator capabilities. After reviewing the data and films and after extensive discussion, it was the consensus of opinion that conservative as the test techniques were, the school training does not teach comparable techniques; and special instruction would be required for students to duplicate the test performance. The dive and flare attitudes were items of particular interest.

33. The prediction method previously discussed allows variables to be changed and the resultant performance calculated. It was the opinion of the pilots at the school that a maximum nose-down attitude of 15 degrees was more representative of actual operating procedure in the event of an engine failure at low speed and high altitude. Since they had not been trained and had not practiced in this area, they were unable to precisely define a pilot's technique for reaction to such conditions. The curves based on the sample calculation are shown in figure B. This is shown primarily to illustrate how technique changes will influence test results and suggest a method whereby the test profile could be modified accordingly.

Autorotational Landing

34. The final portion of the test was performed to demonstrate full autorotative landing capability (both engines inoperative). The tests were conducted at the high-altitude site, and the data were corrected to SL conditions as shown in figure 9, appendix II. The criticality of the maneuver is obvious. It is important to note two aspects: First, the technique stipulated that the collective not be used to cushion the landing. This is the most difficult method out it yields the most realistic data. Second, the touchdown speeds and ground-roll distances were excessive. Thus, for other than an extended paved runway, considerable damage should be expected from an actual emergency landing.

FIGURE 8
PERFORMANCE CHANGES WITH VARIATIONS IN PILOT TECHNIQUE



CONCLUSIONS

35. The following general conclusions were reached as a result of the height-velocity testing of the CH-47B helicopter:
- a. The test specified in reference 6, appendix I, was accomplished by the contractor.
 - b. Analysis techniques can be used to predict height-velocity performance.
 - c. The high forward speeds during autorotational touchdown indicate that in the absence of a prepared landing surface, considerable damage should be expected.
 - d. Extensive prediction and analysis techniques can significantly increase flight test safety and program productivity.
 - e. Stabilization prior to entering a point and accurate knowledge of environment can be critical when conducting H-V tests.
 - f. Real time telemetry monitoring can be of assistance to overall flight safety but cannot be expected to prevent an accident.
 - g. An excessive altimeter position error exists during a high-power OGE hover condition.
 - h. When accompanied by specific pilot techniques the test results are a valid basis for handbook presentation.

RECOMMENDATIONS

36. The following recommendations are made after analysis of the test results and an evaluation of Army aviator capabilities:

- a. The test results should be used to calculate H-V curves suitable for the pilot's handbook.
- b. The pilot technique upon which the calculations are based be detailed in the handbook.

37. The following recommendations should be considered prior to conducting future H-V tests:

- a. Test techniques should be compatible with those normally applied by Army aviators.
- b. Close monitoring of test environment and specific aircraft conditions.
- c. Maximum utilization of engineering analysis prior to test flying.
- d. Establish realistic limits on allowable touchdown speeds and ground distance.

APPENDIX I. REFERENCES

1. Letter, USAAVSCOM, AMSAV-ERS, subject: Boeing/Vertol Proposal for Height-Velocity Testing, CH-47B, 10 April 1968.
2. Letter, USAAVNTA, SAVTE-P, subject: Boeing/Vertol Proposal for Height-Velocity Testing, CH-47B, 30 April 1968.
3. Specification, Vertol Division of The Boeing Company, 114-FT-024-1, Height-Velocity Testing CH-47B Helicopter, 11 September 1968.
4. Specification, Vertol Division of The Boeing Company, 114-PJ-702, Detail Specification for the Model CH-47B Helicopter, 12 April 1966.
5. Report, USAAVNTA, Engineering Flight Test of the YCH-47B Helicopter, Army Preliminary Evaluation I, II, and III, January 1969.
6. Letter, Vertol Division of The Boeing Company, subject: Contract DAA J01-67-A-001(M), DA 23-204-4MC-03397(T); Height-Velocity Testing, CH-47B, 4 March 1968.
7. Specification, Vertol Division of The Boeing Company, 114-PJ-803, Detail Specification for the Model CH-47C Helicopter, 7 December 1967.
8. Report, AFFTC, FTC-TR-66-2, Category II Performance Tests of the CH-47A Helicopter, May 1966.
9. Report, Vertol Division of The Boeing Company, 114-FT-030-1, Analysis of CH-47B Helicopter Height-Velocity Test Program, 20 May 1969.

APPENDIX II. TEST DATA

FIGURE 3

Ground Roll Distance

- NOTES: 1. All landings shown - maximum braking and normal braking.
2. Data base: CH-47B Height-Velocity flight tests.
3. Solid symbols denote A.P.E. testing (minimum ground roll attempted).
4. Sea level test site.
5. (*Denotes minimum ground roll attempted).

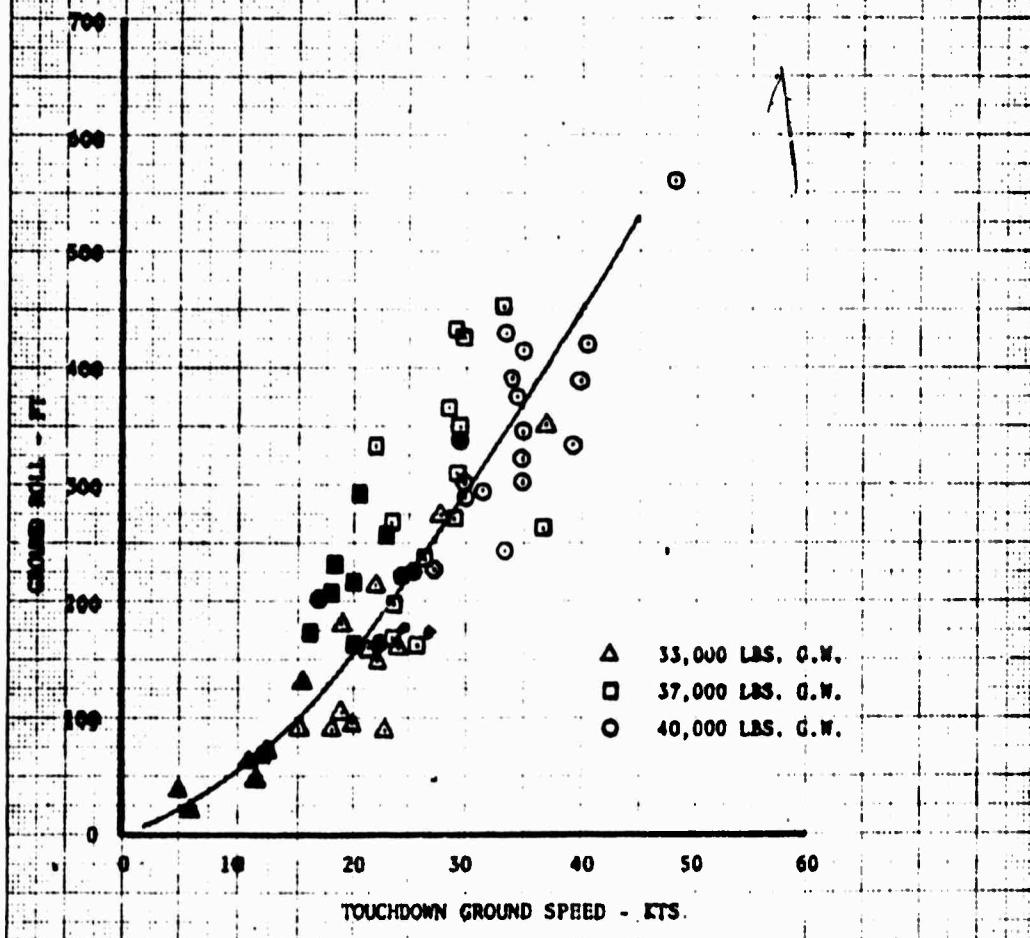
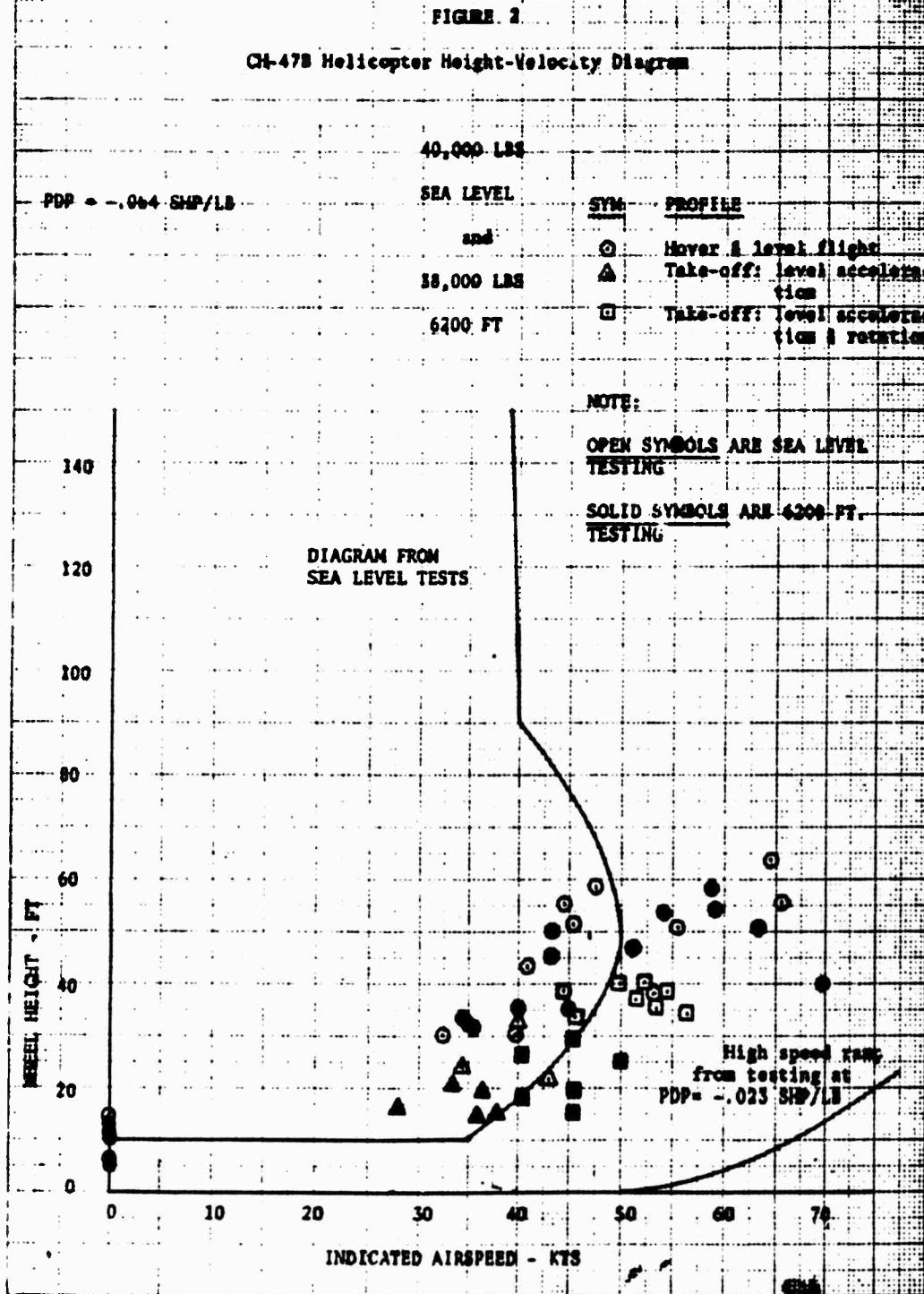


FIGURE 2

CH-47B Helicopter Height-Velocity Diagram



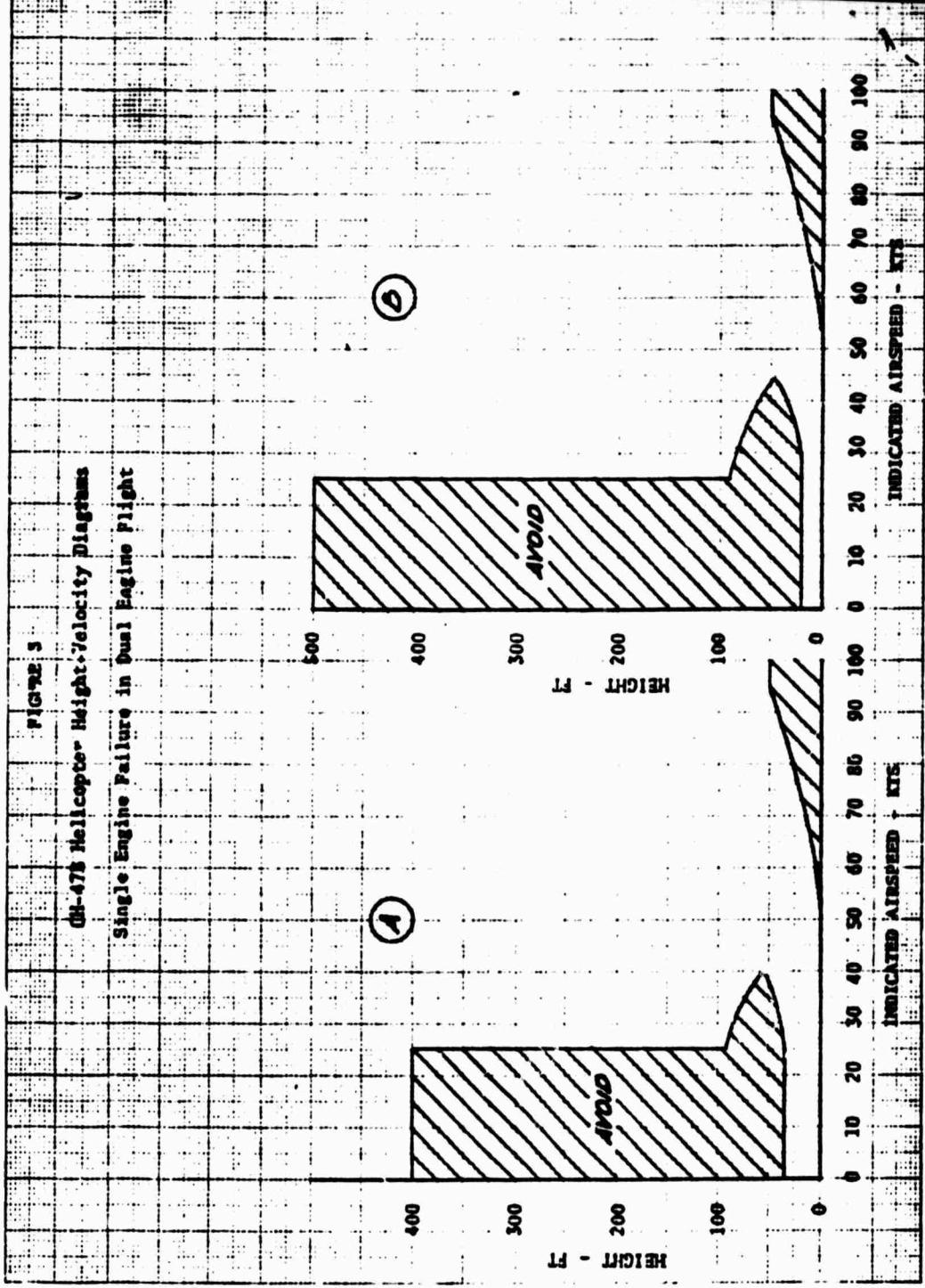


FIGURE 4

CH-47B Helicopter Height-Velocity Diagrams

Single Engine Failure in Dual Engine Flight

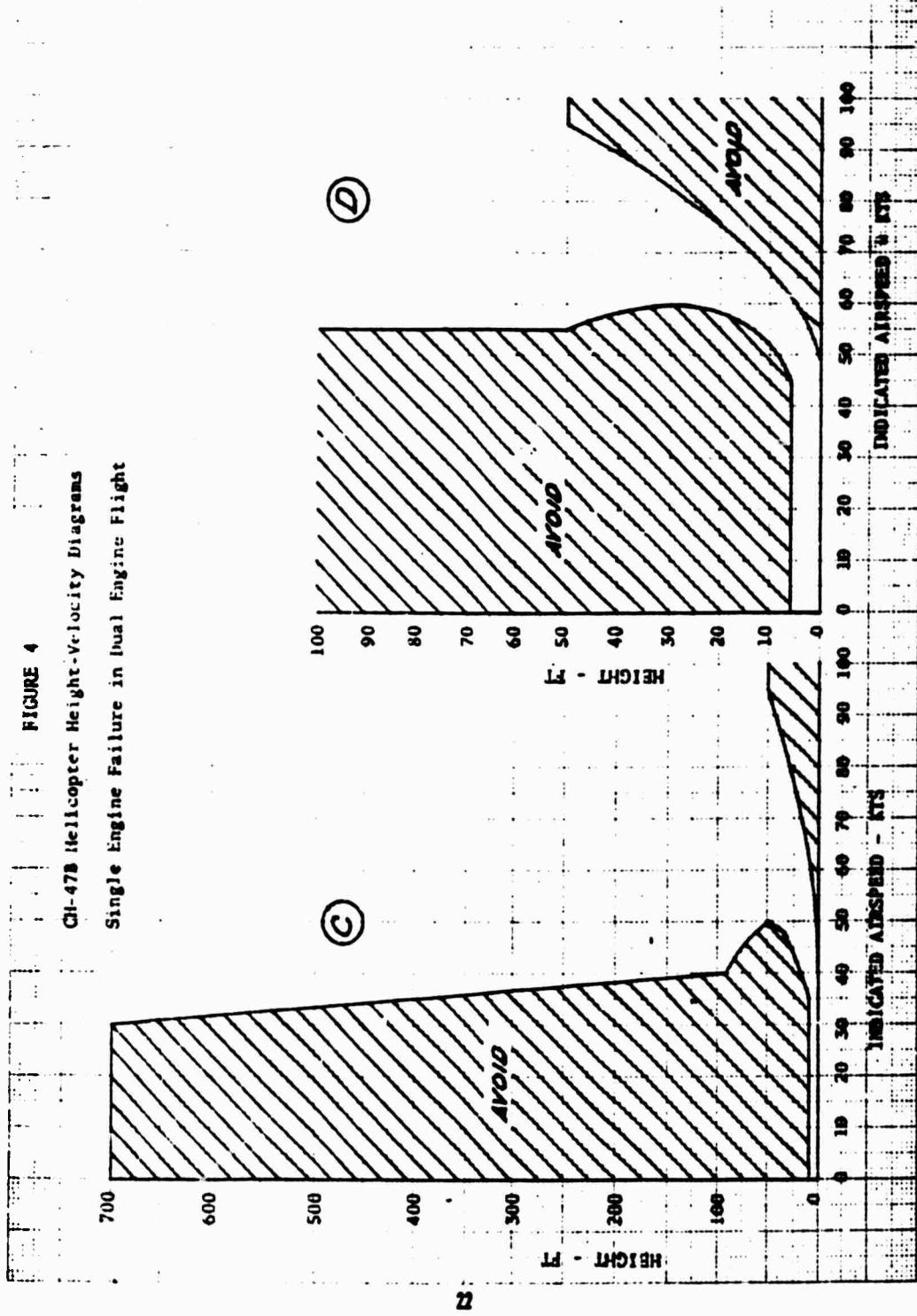


FIGURE 5

CH-47 Chinook Helicopter Gross Weight Limitations

Sea Height-Velocity-Diagram

SEA LEVEL

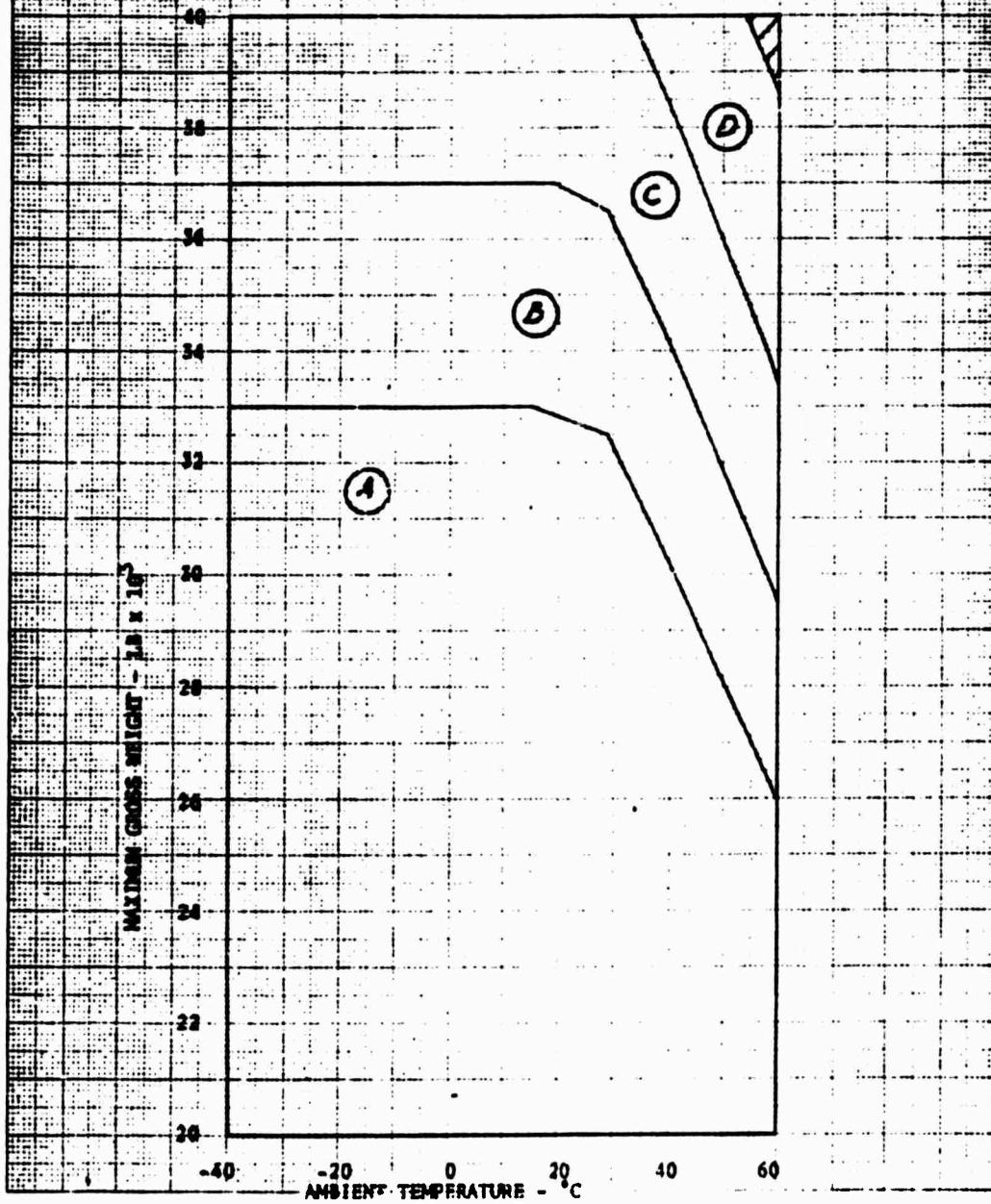
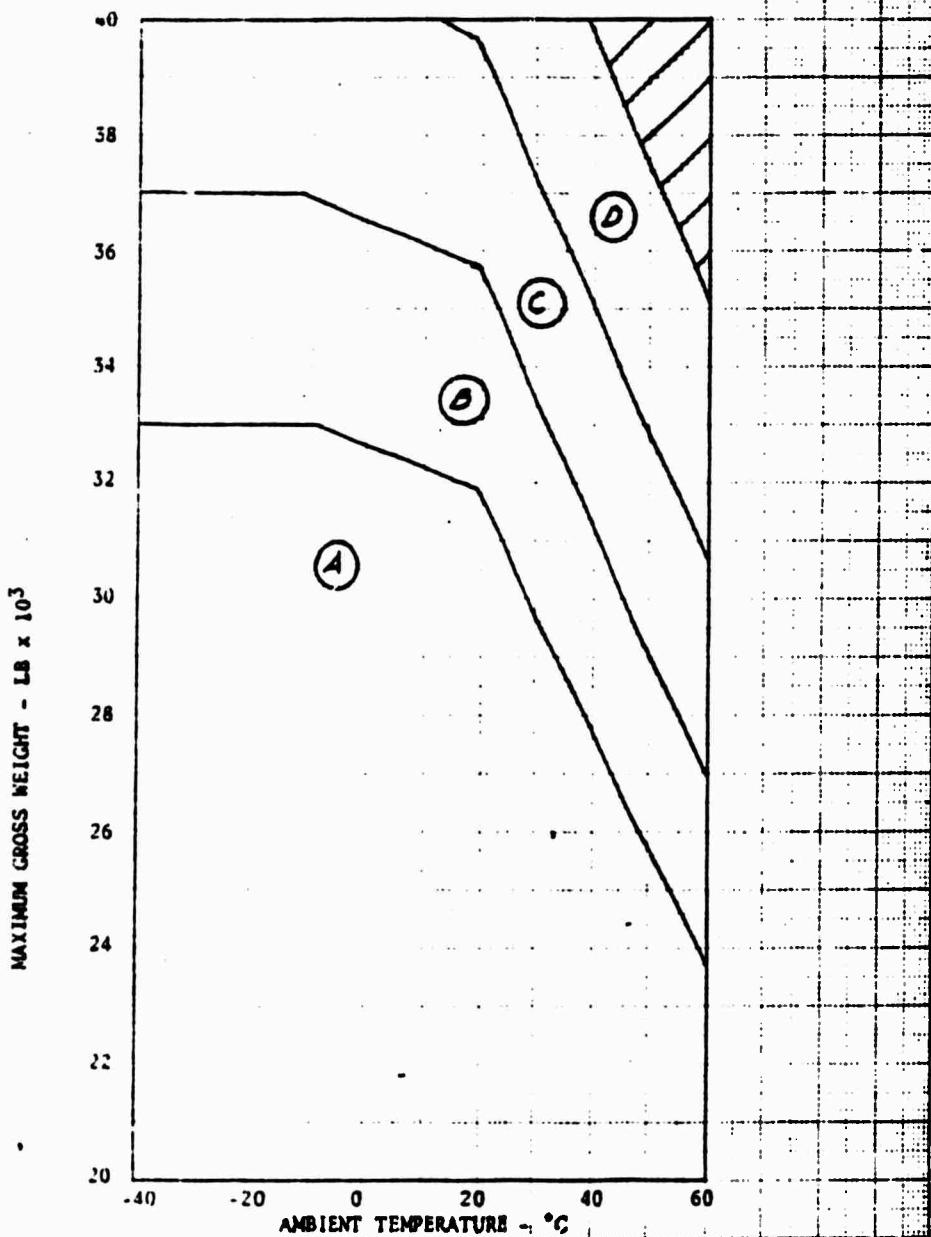


FIGURE 6

CH-47 Chinook Helicopter Gross Weight Limitations

for Height-Velocity Diagram

2,500 FEET



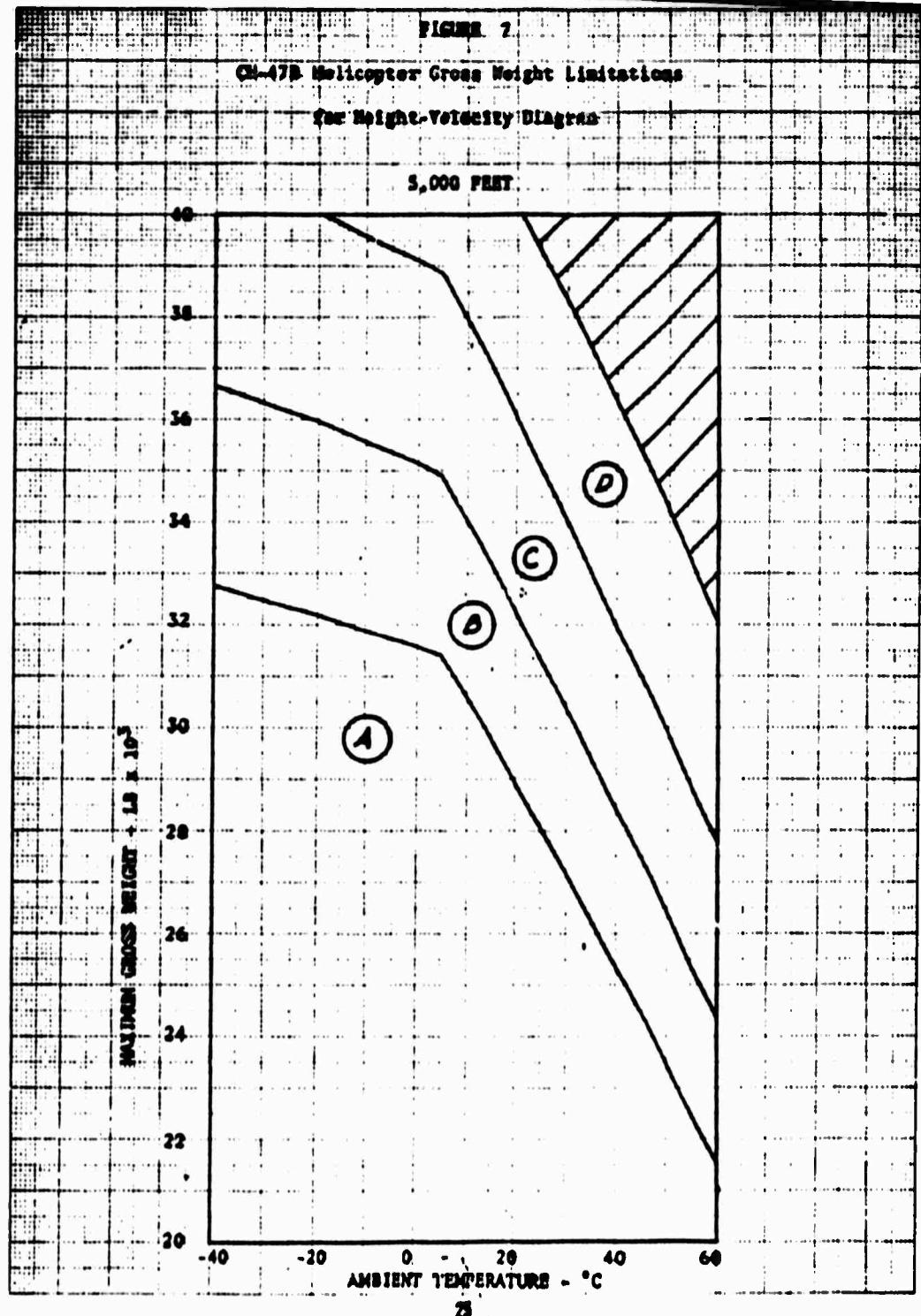


FIGURE 8

CH-47B Helicopter Gross Weight Limitations

for Height-Velocity Diagram

7,500 FEET

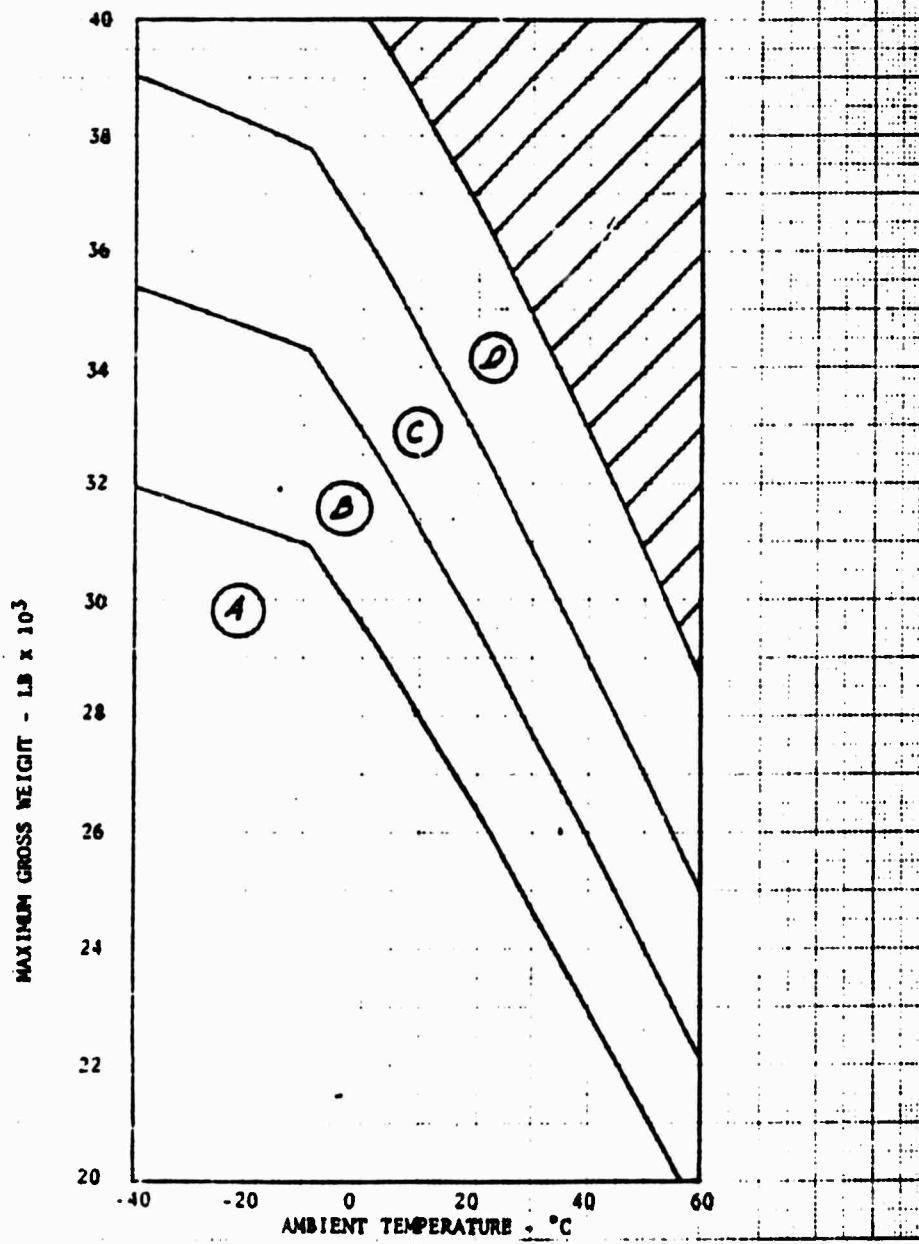


FIGURE 9

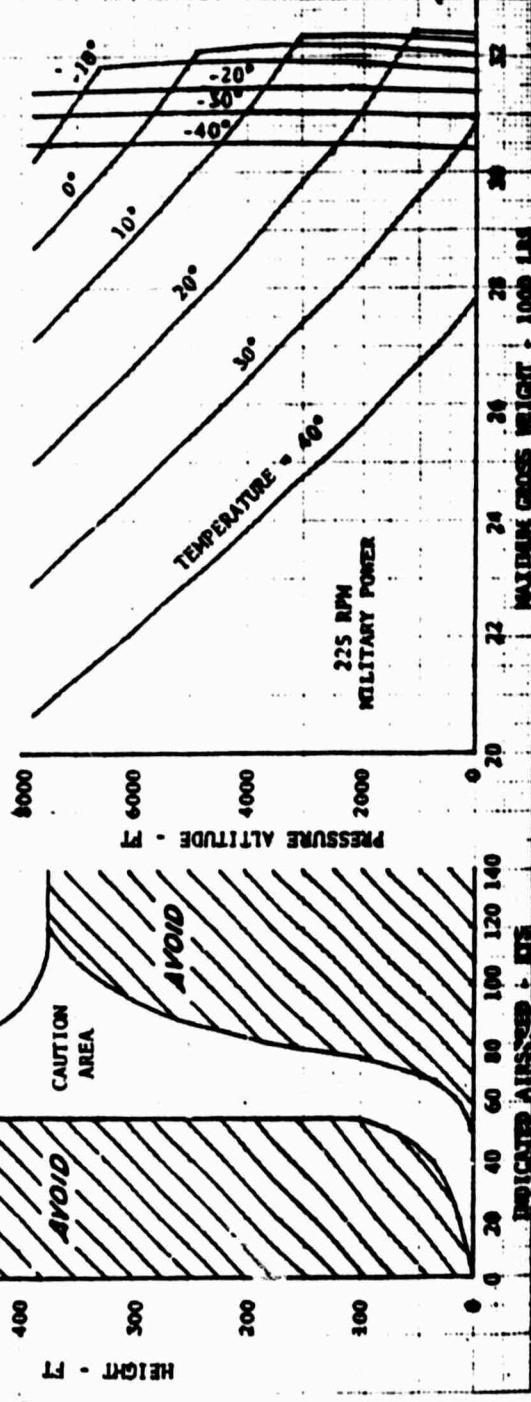
CH-47 Helicopter Autorotation Approach Corridors

Second Engine Failure
From Single Engine
Level Flight

A safe autorotative landing can be made from this line from single engine level flight upon failure of the second engine.

AUTOROTATIVE APPROACHES SHOULD BE FLOWN
WITHIN THE CAUTION AREA

MAXIMUM GROSS WEIGHT FOR
SINGLE ENGINE SERVICE CEILING



1 UNCLASSIFIED

Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) US Army Aviation Systems Test Activity (USASTA) Edwards Air Force Base, California 93523	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED
	2b. GROUP

3. REPORT TITLE

CH-47B HEIGHT-VELOCITY EVALUATION.

4. DATE OF REPORT (Mo. Year) 1st Aug 685. AUTHOR(S)
ALLAN L. HOFFMAN, Cpt, ORDC, US Army, Project Engineer
WAYNE B. DAVIS, Maj, CBT, US Army, Project Pilot
JOHN J. SHAPLEY, JR., Project Officer

(12)-4C

6. PERIOD COVERED Feb 68	7a. TOTAL NO. OF PAGES 1	7b. NO. OF REPS 9
8. CONTRACT OR GRANT NO. 11	8a. ORIGINATOR'S REPORT NUMBER(S) (14) USASTA 68-82	
9. PROJECT NUMBER USAAVSCOM 68-92	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) N/A	

10. DISTRIBUTION STATEMENT

This document may be further distributed by any holder only with specific prior approval obtained through the CG, USAAVSCOM, ATTN:
AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY Commanding General USAAVSCOM, ATTN: AMSAV-R-F PO Box 209, St. Louis, Missouri 63166
-------------------------	---

13. ABSTRACT

The CH-47B height-velocity testing was conducted at Bakersfield and Lake Tahoe, California, during the period 21 October 1968 to 7 February 1969. The program was a joint effort by the US Army Aviation Systems Test Activity (USASTA) and Vertol Division of The Boeing Company (Boeing-Vertol). Boeing-Vertol personnel developed the recommended diagrams which were then evaluated by USASTA personnel. A modified version of the power deficiency parameter method of prediction provides an accuracy similar to that with which the tests can be conducted and data recorded. The resulting height-velocity diagrams were obtained with a specialized technique and may not be suitable for inclusion in the operator's manual. Thus, there is a need to define a suitable pilot technique, make the necessary calculations and accomplish the necessary verification testing.

036470 ✓

DD FORM 1 NOV 68 1473 1473
REPLACES DD FORM 1473, 1 JAN 64, WHICH IS
OBsolete FOR ARMY USE.

UNCLASSIFIED

Security Classification

14. KEY WORDS	LINE A		LINE B		LINE C	
	ROLE	WT	ROLE	WT	ROLE	WT
Ch-47B height-velocity test Power deficiency parameter method Height-velocity diagrams Specialized technique Not suitable for operator's manual Verification testing required						